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DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

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## ABSTRACT

This document was prepared for presentation to the 24th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Melbourne, Australia on May 1 and 2, 1995.

A review is given of the aircraft fatigue research and associated activities which form part of the programs of the Aeronautical and Maritime Research Laboratory, Universities, the Civil Aviation Authority and the Defence Scientific Establishment, New Zealand. The review summarises fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.

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**CONTENTS**

## 9.1 INTRODUCTION

This review of Australian and New Zealand work in fields relating to aeronautical fatigue in the 1993 to 1995 biennium comprises inputs from the organisations listed below. The authors acknowledge these contributions with appreciation. Enquiries should be addressed to the person identified against the item of interest.

| | |
|---|---|
| AMRL | Aeronautical & Maritime Research Laboratory, GPO Box 4331, Melbourne, Victoria 3001. |
| CAA | Civil Aviation Authority, PO Box 367, Canberra, ACT 2601. |
| CRC-AS | Cooperative Research Centre - Aerospace Structures, 506 Lorimer St., Fishermens Bend, Victoria 3207. |
| DSE | Defence Scientific Establishment, Auckland Naval Base, Auckland, New Zealand. |
| MU | Monash University, Wellington Road, Clayton, Victoria 3168. |
| RMIT | Royal Melbourne Institute of Technology, GPO Box 2476V, Melbourne, Victoria 3001. |
| U of S | University of Sydney, New South Wales 2006. |

## 9.2 FATIGUE PROGRAMS ON MILITARY AIRCRAFT

### 9.2.1 F/A-18 Fatigue Investigations

#### 9.2.1.1 F/A-18 Operational Loads Monitoring (L. Molent - AMRL)

The design specification for the F/A-18 required that the aircraft be designed to achieve a 6000 hour (safe) life of type to a predefined manoeuvre and operational environment. This environment included manoeuvres, carrier landings, field landings, catapult launches and touch-and-go landings, which were anticipated as typical for operation of such an aircraft type (fighter/attack) in the United States Navy (USN) environment. To assist the fleet manager in achieving the required life of type the design authority (Naval Air Systems Command - NAVAIR) and the prime contractor (or manufacturer - McDonnell Douglas Aerospace - MDA) developed a system of fatigue life monitoring tools. This system has been adapted for use by the Royal Australian Air Force and is known as the F/A-18 Maintenance Data and Service Life Monitoring System (MD&SLMS). The MD&SLMS includes the aircraft on-board data acquisition system and analysis software derived from the NAVAIR F/A-18 Structural Appraisal of Fatigue Effects (SAFE) program. The basis of this program is the aircraft's on-board data acquisition system known as the F/A-18 Maintenance Signal Data Recording System (MSDRS). The MSDRS is an omnibus system integrated with the aircraft avionics and flight control system which records time based data from the aircraft's data base, including maintenance, engineering, flight incident, tactical data and data from strain sensors located at seven fatigue critical locations. This system provides the basic structure for operational load monitoring for all operators of the F/A-18 aircraft.

The USN approach to fatigue life management is fundamentally different to the durability and damage tolerance analysis (DADTA) approach generally adopted by the RAAF, and as such the RAAF perceive a number of difficulties with the USN approach, the most obvious of these being the differences between RAAF operations and the F/A-18 design environment, and that the inspectability and maintainability of components have not been fully assessed. To address some of these differences, the RAAF, in partnership with the Canadian Forces, are conducting the International Follow-On Structural Test Project, Section 9.2.1.3, to resolve the deficiencies of the specification testing, and has introduced additional fatigue life management tools such as the Aircraft Fatigue Data Analysis System (AFDAS), [1,2,3,4].

The AMRL-developed AFDAS collects and processes aircraft strain and accelerometer sensor data. These AFDAS sensors provide additional monitoring capability to the MSDRS sensors. Of particular importance is its capability to monitor the strain distribution across the three wing attachment bulkheads. The AFDAS was intended to provide a rapid means of assessing airframe fatigue usage as the collected data are compressed on-board the aircraft and thus requires less processing than does the MSDRS data. (AFDAS is also fitted to a sample of aircraft in each of the F-111, PC-9 and Macchi fleets.)

The AFDAS has the potential to provide valuable fatigue usage information that is not available through the MSDRS, and may provide a capability for simple data extraction and fatigue damage assessment at the aircraft squadron level. Until recently, little progress had been made in achieving these goals, primarily because all effort has been devoted to developing and validating a system based on data obtained from the MSDRS. A substantial effort is now underway, however, to enable mature AFDAS data collection. As the function of the MSDRS is well documented elsewhere, the next section will concentrate on the AFDAS.

#### 9.2.1.2 Aircraft Fatigue Data Analysis System (AFDAS) (L. Molent - AMRL)

The AFDAS was pioneered by AMRL and was developed and is currently marketed by British Aerospace Australia Limited. It directly monitors and records relevant fatigue loads data occurring at a number of selected locations on the F/A-18 aircraft structure. The locations of some AFDAS strain sensors are mirrored by MSDRS sensors. A further channel records aircraft normal acceleration from the aircraft's own backup *cg* accelerometer.

Central to the AFDAS is the airborne Strain Range Pair Counter (SRPC), which automatically processes and stores the information from the sensors. Fresh data are added to the data base each time the aircraft flies. These data can then be transferred to a useable medium (e.g. floppy disc) by means of a portable data readout computer. This is normally conducted on a monthly basis. Software is available to interrogate these data to provide additional fleet management capability. The SRPC monitors the output of the sensors during the flight load-time waveform, and pairs and then extracts the range-pairs, which form the basis of subsequent analyses.

*AFDAS Operational Status*

Each of the RAAF's F/A-18 was equipped with AFDAS during manufacture. Data have been collected from the fleet since early 1992. Due to other priorities, the data have only recently been interrogated for integrity, and numerous problems have been encountered. Steps are now underway to remedy these. Numerous hardware problems and inoperative sensors have been detected by the AFDAS screening program, which was developed by Hawker De Havilland Victoria.

Although the AFDAS contains an "end of flight" (EOF) marker, this must be deployed manually. Thus the data collected to date are not in flight-by-flight format, as the EOF marker is deployed only at the time of data extraction (i.e. monthly). A modification contract has been let to the manufacturer to integrate the EOF marker with the aircraft "weight off wheels" sensor, which will then enable the collection of flight-by-flight data. This will greatly aid the interpretation of the usage data, and will simplify correlation with MSDRS data.

Other areas of progress include:

Validation of the Preselected AFDAS Sensor Ranges, and Transfer Function Derivation—By using data available from the ST-16 fatigue test, IFOSTP phase I flight trials and AFDAS data collected from aircraft A21-034, it has been shown that the strain ranges for the AFDAS sensors originally specified were not optimal. New ranges have been defined to ensure that useful AFDAS data can be collected, once modifications to the existing AFDAS are made. From these data sources, direct relationships between the AFDAS and MSDRS sensors and applied loading were also derived.

Investigation of Engineering Changes—Production changes for the F/A-18 have been investigated to examine the effect on AFDAS (and MSDRS) response, and it is considered that in general the response of the sensors should not be significantly affected by these changes. During the review several configuration changes were identified which may have significance in F/A-18 fleet fatigue life management.

*Buffet Response of AFDAS Sensors*

The F/A-18 is an extremely versatile, highly agile, high performance fighter/attack aircraft. Its design, with leading edge extensions (LEXs) on the inner wing, provides fore body lift enabling the aircraft to fly at very high sustainable angles of attack (AoA), in excess of 60°. The twin vertical fins canted slightly outboard provide good directional stability during high AoA flying. These flying qualities are enhanced by vortices generated at the front of each LEX, which keep the airflow attached to the wing at high AoA, and provide high energy airflow past the verticals thus enhancing lateral control. During high AoA manoeuvring the vortex flow breaks down or "bursts" into turbulent flows forward of the vertical tails. This produces oscillatory dynamic pressures and buffet loading on the structure. This phenomenon is well documented for the empennage of the F/A-18, and produces large dynamic stresses in the structure. These large dynamic stresses and exposure times at high AoA flight conditions have been shown to contribute to the fatigue and premature failure of the structure, and are the primary reasons for the conduct of the AMRL portion of IFOSTP.

AMRL has recently investigated the occurrence of buffet at structural locations other than the empennage and aft fuselage, which may influence the response of the fatigue monitoring (AFDAS or MSDRS) strain sensors. A preliminary estimate of the structural significance of this buffet response was also made. Data recorded at a high sample rate (approximately 240 Hz) during recent flights trials were used to perform the investigation. It was shown that, apart from the empennage sensors, two other sensor locations are influenced by buffet, namely the outer wing fold and trailing edge flap. Parameters such as dynamic pressure, high angles of attack, high normal accelerations, asymmetric manoeuvres involving high roll rates, heavy configurations points-in-the-sky (PITS) and amount of stores carried were among the quantities examined to determine what flight characteristics produced the buffet noted at the wing fold and the trailing edge flap.

From this investigation it was clear that the onset of outer wing buffet was a function of AoA. No other parameter was found to strongly correlate with buffet onset or the magnitude of oscillation, including dynamic pressure. It was clear that when AoA reached 10°, buffet initiated, and when AoA dropped below 10°, buffet terminated. Configuration and PITS were not seen to influence the onset or magnitude of the buffet. The buffet was observed to occur simultaneously on both port and starboard wings. This result is similar to that previously noted for the empennage of the F/A-18, in that buffet excitation occurs at AoAs greater than 10°. For these cases it can be concluded that wing buffet occurs simultaneously with empennage buffet.

It was clear from the FFT analyses conducted that the wing was responding at its natural frequencies, and good agreement was achieved between those calculated values and those of MDA. In the case of the wing fold, the amplitude of the buffet was determined to be approximately 10% of the maximum expected strain range of the fatigue sensor located at the wing fold. It is considered that this amplitude, coupled with the frequency of flight above 10° angle of attack, may degrade the fatigue life of the structure, and thus warrants additional investigation.

*AFDAS Channel Data Correlation*

AFDAS processing software includes routines to screen the data for potential errors including the following:

a. Documentary data discrepancies including invalid tail number and dates/times.

b. Errors in hardware including amplifier errors, battery voltage low and strain gauge errors.

c. Checking the range pair data outputs and providing warnings if there are counts in the extreme windows, if there are invalid range pair data structure (trough higher than a peak), or if the counts in any window exceed a certain predetermined value.

The data screening as described above is useful and necessary, but is very limited.

A method of checking the correlation of separate channels on the system both internally and against some expected value was identified as being required.

A method of correlating two channels from AFDAS when the data are presented in the form of two range pair tables has been identified. The solution is based on comparing the frequency distributions from two different sources to check for correlation. By comparing the two distributions, it is possible to determine if a linear relationship (as expected) exists between the two. The method has been incorporated into a computer program, and will be used to determine the validity of collected data.

*Upgraded AFDAS Capability*

A modification to the existing AFDAS SRPC to enable the recording of time tagged turning points is considered by the RAAF as highly desirable enhancement to its capabilities. A study by the manufacturer has indicated that this will be possible with relatively few changes of the existing hardware, albeit for a restricted, yet acceptable, recording time. This upgrade will proceed

in the near future, and when available, limited units can be rotated through the fleet in order to obtain time correlated data, which can be used for individual aircraft sensor calibration amongst other purposes.

#### 9.2.1.3 AMRL F/A-18 International Follow-on Fatigue Test Project [IFOSTP] (A.D. Graham and G.W. Revill - AMRL)

Progress on this task has been reported to recent ICAF Conferences, eg [5]. The construction of the IFOSTP test rig and the tuning of the control system has now been completed. The rig has successfully applied a short test sequence of combined manoeuvre and buffet loading. Current testing indicates that the rig will apply the manoeuvre loads at a cycling rate of about 0.75 Hz. The final development phase of the complete test system is about to begin. This is a commissioning and acceptance trial consisting of the application of a full block of loading to the dummy test article. The dummy test article is on loan from the USN for test rig development.

The test article, FT 46, is a centre and aft fuselage and empennage built by Northrop to a standard representative of the RAAF fleet of F/A-18s. Instead of the forward fuselage there is a steel frame to provide a forward reaction point. The test section is all structure aft of and including the 557 bulkhead which is near the forward edge of the vertical fins.

The IFOSTP test system is designed to apply both low frequency (about 0.5 Hz) manoeuvre loading and correlated high frequency dynamic loading (>10 Hz) The high frequency loading represents loads caused by the excitation of structural resonances. In flight these resonances are excited by aerodynamic effects. In the IFOSTP test system the manoeuvre loads are applied by highly compliant pneumatic actuators, and representative modes of vibration are excited by high frequency electromagnetic actuators.

The pneumatic actuators have been developed at AMRL specifically for this test. These are of two types. The first type is a slightly modified Firestone air spring and is used in locations where mass and compliance are less important and the range of dynamic movement is small. The second type is based on commercially available rolling sleeve air springs but made of much thinner and lighter material in order to operate at very low pressure and to minimise damping and mass for application where the dynamic movement is greatest. These actuators will accommodate a stroke of up to 400 mm and apply forces up to 19 kN. To increase their compliance the actuators incorporate reservoirs of up to 0.11 cubic metre. The airflow to the actuators is controlled by 50 mm pneumatic spool valves that were designed and developed by AMRL. The pneumatic power is supplied by two 110 kW compressors feeding two 11 cubic metre reservoirs through a refrigerated air drier.

The high frequency loads are applied by eight electromagnetic actuators, six acting on the empennage surfaces and two on the engines. The actuators can apply a maximum force of 22 kN over a stroke of 115 mm for a maximum peak power of 60 kW. The actuators, which weigh one tonne each, are isolated from the test frame by a one tonne seismic mass mounted on airsprings. Force from the actuators is transmitted to the test article by links with a zero backlash spherical joint at each end. The links are strain gauged to allow measurement and recording of the applied forces. The system is controlled by a multi-exciter vibration control system made by STI which uses the dynamic characteristic of the structure to determine the dynamic forces that have to be applied to achieve the required responses at the tips of the vertical fins and stabilators and the engines.

During the test the test article will deflect by up to 140 mm from the unloaded condition. Other parts of the loading system are not tolerant to such large displacements so the test article is suspended on an active reaction system that largely compensates for the deflection. To do this the reaction points at the wing pickup bulkheads and the nose are mounted on spherical bearings. The fittings at the wing pickup points are then suspended from three hydraulic actuators. These actuators are then controlled so that during the test loading they will maintain selected points on the aft fuselage to within 10 mm of their static position.

The loads to be applied are optimised to represent the service use in both the RAAF and the CF fleets. To determine the loads several flight test and wind tunnel programs have been conducted in both Australia and Canada. The characteristic usage has been estimated from extensive fleet surveys and fleet monitoring programs in both countries.

The test system is controlled by a digital control system designed and developed at AMRL. This system is capable of controlling up to 256 channels although only 65 are used on the IFOSTP rig.

Because of the high frequency of the dynamic loads a high rate of data acquisition is required to enable the strain response of the structure to be measured and recorded. A Hewlett Packard data acquisition system is being used to record 112 channels at 512 Hz and 490 channels when the load is constant at load line dwell points. The raw data are expected to amount to about 1.5 terabytes of information all of which is to be archived on CD-ROMs before it is distributed to users in reduced forms.

Test article inspections will use a variety of methods including X-ray, C-Scan, acoustic emission and LFECA to provide comparative effectiveness to assist in planning fleet inspection methods. The test article has its own insertion/extraction rail system to allow it to be removed from the test rig for inspection. Detailed inspection of previous test articles has been difficult because of the large number of loading system components attached to the test article. Inspections are initially planned at the end of each block of loading but the inspection frequency will be adjusted in the light of experience as the test proceeds, and in line with fleet inspections.

The final commissioning is expected to start soon and will be in parallel with an external design review of the total test system. When the commissioning is complete the dummy test article will be replaced with FT46. Continuous test running is planned to begin in July 95 after a series of strain surveys and the dynamic characterisation is completed.

#### 9.2.1.4 F/A-18 Fatigue Life Management (L. Molent - AMRL)

In 1987, a review of the RAAF F/A-18 Maintenance Data and Service Life Monitoring System (MD&SLMS) indicated that the fleet usage severity was in excess of the design usage rate, indicating that the expected life-of-type may not be achieved. Several usage management plans were implemented, including the release of the Mission Severity Monitoring Program (MSMP), which is a very simplified version of the SAFE software, Section 9.2.1.1, which enables individual squadrons to calculate usage rates on a flight-by-flight or mission basis. Periodic assessment of the MSMP usage rates indicate similar trends to that given by the MD&SLMS. The RAAF usage management has been so successful that recently the management limits have been significantly relaxed such that the usage severity can be allowed to increase without detriment to the required aircraft fatigue life.

#### 9.2.1.5 Assessment of Cracking in F/A-18 Bulkhead Fatigue Test (G Clark - AMRL)

The assessment of fatigue cracking occurring in a full-scale fatigue test of an FS488 bulkhead from an F/A-18 aircraft was given in the previous Review [5]. In summary, the multitude of small cracks at several locations was testimony to the efficacy of the design, and a necessity for safe life fatigue management of this component—safety-by-inspection will probably not be an option here.

The cracking behaviour seen on this test—small multisite cracking—carries with it an increased sensitivity to machining/manufacturing defects, in-service damage, corrosion or metallurgical defects. A substantial test program is underway to quantify the effect of glass bead peening on the fatigue life of aluminium alloys. The results indicate that the effect of the surface damage introduced by peening needs to be considered very carefully: in some cases, peening can reduce the life of the component. Tight control of the peening conditions, however, has been shown to provide a substantial life improvement, and the possibility of a rework procedure is being evaluated. Such a procedure is based on removing peened and cracked surface material, followed by re-peening, [6].

#### 9.2.1.6 Cracking in F/A-18 Trailing Edge Flap Hinge Lugs (G. Clark - AMRL)

Investigation of the loss of an F/A-18 trailing edge flap (TAF) in flight led to the conclusion that a hinge lug had fractured by fatigue. The small critical crack size and difficult NDI configuration led to development of a supplementary NDI procedure which would detect small defects. Inspection of the fleet revealed the presence of further cracking. In all cases, the cracking was associated with the presence of corrosion defects. The components are being managed on a safety-by-inspection basis, pending redesign, and with improved corrosion protection schemes, [7].

#### 9.2.1.7 F/A-18 Coupon Fatigue Tests (J.M. Finney - AMRL)

Six fatigue-critical regions have been identified on the F/A-18 rear fuselage and coupon fatigue tests are planned to simulate each of these regions. Most of the work to date has concentrated on the vertical tail stub frame region and the horizontal stabilator spindle, and aims to compare the fatigue behaviour of Canadian, Australian, and any compromise sequences. It is clear that buffet loads dominate the fatigue life for the vertical fin, but for the stabilator spindle the buffet and manoeuvre loads are equally damaging. Fractographic analysis of the coupon data is being undertaken in order to ensure fractographic readability of cracking in the full-scale test article.

Fatigue life predictions for all test sequences are being made and compared with experimental lives. The main reason for these comparisons is to gain confidence in the use of life predictions which will assume some importance once a full-scale article failure has been obtained. Both crack initiation and crack growth models are being used, details of which were given in the previous review, [5].

Basic fatigue data (e.g. da/dN–$\Delta K$) are also being obtained for relevant F/A-18 materials in order to validate the sets in current use.

#### 9.2.1.8 Engine Life Usage and Durability Enhancement (N.S. Swansson - AMRL)

Prompted in part by fretting fatigue and other LCF problems in rotor components of the F404 engine fitted to F/A-18 aircraft, the (approximate) life usage algorithms used for maintenance life management are being compared with life estimates determined from flight history data for the engines. In a similar exercise conducted with data obtained in 1986, optimising the lifing algorithms for Australian operations produced useful extensions of life for some components.

Concurrently a comparison was made between 1986 and 1993 on the severity of engine usage and its effects on expected fatigue lives. Overall counteracting changes were observed. Average flight time increased, thereby reducing the damage rate due to major fatigue cycles and increasing expected life. But with greater familiarity pilots manipulated power lever angles more often so using the engines more vigorously, increasing the damage rate from minor cycle usage and reducing life. In total the net changes in expected life were small.

Durability enhancement is concerned with thermo-mechanical fatigue of engine hot section components. Present activities have two main thrusts:

a) Determination of temperature loads in hot-section components. This includes both application of advanced temperature measurement techniques, and development of expertise in turbine heat transfer in order to predict metal temperatures in components which are heated by combustion products (also radiation in combustors) and cooled by compressor bleed air.

b) Structural analysis of components subject to thermo-mechanical fatigue loading. Finite element systems have been applied, using both traditional creep-plasticity models for materials, and the more recently developed state variable constitutive models. Little has been published regarding the use of constitutive models under the non-isothermal conditions of thermo-mechanical fatigue.

Work supported by AMRL has led to improved convergence and solution efficiency when the Ramaswamy-Stouffer-Bodner model is implemented in the ABAQUS finite element system. It has also revealed that with commonly used integration schemes plus an approximate Jacobian matrix, the solution may converge to an erroneous value. The latter problem is not manifest in typical low temperature components where inelastic deformation is localised in regions of high stress concentration. Work is continuing to formulate a revised integrator cum revised Jacobian which will converge correctly in high temperature components where inelastic deformation is extensive.

### 9.2.2 F-111 Fatigue Investigations

#### 9.2.2.1 F-111C DADTA Support (K. Watters & J. Paul - AMRL)

The RAAF manages its fleet of F-111C aircraft on a safety-by-inspection basis complemented by cold proof load tests (CPLTs). Safe inspection periods are derived from durability and damage tolerance analyses (DADTAs) performed by the Lockheed Fort-

Worth Company (LFWC) specifically for the RAAF. Major inputs to the DADTAs are the stresses at the identified critical locations around the aircraft structure.

Two of these locations are presenting most concern to the RAAF for structural integrity of the aircraft. They are known as stiffener runout number 2 (SRO#2) and fuel flow vent hole number 13 (FFVH#13). Both are in primary structure of the wing pivot fitting (WPF) and have produced numerous incidences of cracking detected in the fleet. At the same time the stresses at these locations are very complex, and the cracking is believed to be driven by residual tensile stresses caused by localised yielding during CPLT. AMRL was tasked by the RAAF to analyse the cyclic and residual stresses at SRO#2 and FFVH#13 for input to DADTAs at these locations.

Preliminary investigations showed that classical plasticity theory cannot realistically represent the cyclic plasticity occurring at SRO#2 and FFVH#13. AMRL adopted the Stouffer-Bodner constitutive model, [8], for implementation in the PAFEC FE package to track the cyclic plastic behaviour and accurately predict the residual stresses produced by CPLT.

SRO#2 was analysed during 1993/94 and the results sent to LFWC who performed a DADTA which produced workable inspection intervals for the RAAF, approximately double the length of the previous inspection intervals. The SRO#2 stress analysis was done for a matrix of parameter values representing the range of geometric configuration states of the fleet. The more important parameters were the presence or not of WPF reinforcement doublers and the height and radius of the runout which are affected by reworks to remove cracks. The SRO#2 stress analysis was actually a hybrid process with the FE solution performed using classical plasticity and the stresses derived from the strain field using the Stouffer-Bodner model.

AMRL is currently working on the FFVH#13 stress analysis. The Stouffer-Bodner constitutive model has now been implemented into the PAFEC code to obtain a direct solution. This model and its PAFEC implementation are being validated against well defined representative specimen tests. An FE model of FFVH#13, allowing for a number of configuration shapes representative of the range of fleet rework shapes, is being developed.

A full-scale static test program to simulate the CPLT loading of an F-111C wing is in progress. The rig has been modified to allow for a range of wing sweep angles to be tested. The wing pivot fitting of the test wing has been instrumented with approximately 600 strain gauges, including extensive gauging of the shear ring which is an area of particular concern to the USAF because of cracking in its fleet. Modification of SRO#2 and FFVH#13 in the test wing will be done between runs to gather data from rework shapes corresponding to those of the FE models. The strain data will be used to calibrate the AMRL FFVH#13 FE model. They will also be used as part of a collaborative agreement with the USAF to calibrate an FE model of the WPF being developed by LFWC.

#### 9.2.2.2 F-111 AFDAS (K. Walker - AMRL)

The Aircraft Fatigue Data Analysis System (AFDAS) is a strain based fatigue data collection and analysis system. The system consists of a central processor and recorder with strain gauge sensors placed at fatigue critical locations on the structure. The current version of the system (Mark III) records data from eleven strain channels and one cg vertical acceleration channel. The sensor signals are processed according to a range-mean-pair counting algorithm and the counts are stored in a 120 cell range-mean-pair table.

Rather than storing the strain information as a direct time based history, the AFDAS system processes the sensor signals in real time and quantises the peaks and troughs into pre-defined levels or bands. The peaks and troughs are matched according to a range-mean-pair counting algorithm and are stored on a flight-by-flight basis as counts or occurrences of a particular peak/trough (or mean/amplitude) combination.

The RAAF has recognised the potential benefits of the AFDAS system in providing an accurate and comprehensive source of data for fatigue monitoring purposes. Recording strains directly at the point of interest is seen as a significant improvement on fatigue meter and parametric based techniques. AFDAS has been implemented on a number of RAAF aircraft types including F/A-18, PC-9, Macchi and F-111.

The application of AFDAS to the F-111 was initiated in the early 1980s. The installation consists of four strain gauges fitted to each wing, seven fuselage gauges and one cg vertical acceleration sensor. Both wings are modified, but only the left wing gauges are normally connected. The equivalent gauges in the right wing can be connected in the event of gauge failure. The system is to be installed on eleven F-111 aircraft, and the gauges are to be installed on all wings to allow for wing movements during maintenance. The current status is that the modification has been incorporated on seven aircraft, one aircraft is being modified and the final three await modification.

The modified aircraft in active service are now producing data. The data are transferred to disk once per month per aircraft, which is then forwarded to the RAAF Aircraft Structural Integrity Section and to AMRL for analysis and advice. The results so far have indicated some problems with the data quality and integrity. Further analysis and development is required to advance the status of the system to the point where it provides consistent, useable and relevant data to assist in the structural life management process.

Achievements to date are as follows:

a. Hardware and software errors have been identified and corrective actions initiated. The effectiveness of these measures will be evaluated when new data are collected.

b. An inter-channel correlation procedure has been developed and tested. This procedure has been incorporated into a computer program and will be used to assist with data screening and quality checks.

c. The ability to compare AFDAS-generated spectra directly with spectra generated from parametric data has been demonstrated. This is achieved by ensuring that both spectra are represented in either range-mean-pair table or exceedence plot format.

d. Significant progress has been made in determining transfer functions to relate strain at the AFDAS gauge locations to stress at relevant control points.

#### 9.2.2.3 Analysis of an Interference-Fit Plug Option for Life Extension of the F-111C Fuel Flow Vent Hole Number 13 (M. Heller - AMRL)

A significant structural integrity problem for the fleet of F-111 aircraft in service with the RAAF is fatigue cracking at fuel flow vent hole number 13 in the wing pivot fitting, Section 9.2.2.1. Currently, this problem is being managed by reworking the hole to one of a family of progressively larger shapes, in order to remove any small cracks or corrosion when detected. These optimal (non-circular) shapes were previously determined by analysis at AMRL. In practice the reworking does not, however, completely eliminate further cracking, and at the current rate of crack growth and reworking, this procedure may not enable the aircraft to reach its desired life-of-type.

AMRL is now developing a repair option for the fuel vent hole to eliminate or significantly reduce the crack growth rate in this region. The repair approach involves standardising the hole shape throughout the fleet, and then installing interference-fit plugs. Obtaining a viable solution to this problem is complicated by the fact that the hole is *non-circular*, and access to the hole for machining purposes is difficult.

Coupon experimental stress analysis and finite element analysis of the repair option has progressed well. It has been shown that the alternating stress levels are significantly reduced by interference-fitting a suitable plug into the hole. Furthermore, the results obtained indicate that the level of interference stresses, alternating stresses and full contact hole boundary conditions can be optimised to achieve the desired solution.

Analytical and experimental work is currently underway to investigate the benefits and viability of hole cold-expansion prior to interference-fitting the plug.

#### 9.2.2.4 F-111 Lower Wing Skin Repair (L.R.F. Rose - AMRL)

Results obtained to date as part of a comprehensive repair-substantiation program for a bonded repair to a cracked wing skin are given in [9]. The design methodology relies on identifying an appropriate bonded joint, representative of the load transfer across the crack. This can be used both to simplify the stress analysis required for designing the repair and, more importantly, to generate the relevant design variables. A three-dimensional finite-element model, validated against a strain survey on a full-scale test wing, is used to quantify the stress concentration factor and secondary-bending effects due to geometrical details around the site of cracking. The implications of these results for the design calculations and for the validation testing are briefly discussed in [9].

#### 9.2.2.5 F-111 Ultra-High-Strength Steel Machining (G. Clark - AMRL)

A safety-by-inspection approach is being used to manage some fatigue-critical locations in the high-strength steel wing structure in RAAF F-111Cs. While the non-destructive inspections required are very sensitive, the logistical implications of the testing are such that the area is also subjected to a confidence cut to ensure that cracking which is below the NDI limit is removed. The difficulties in controlling fine machining in the fatigue critical regions of high-strength steel may be resolved by an improved electro-discharge machining method which is capable of leaving only a very shallow heat-affected zone. Such a zone is easily removed mechanically, and it can be easily detected if not removed. The approach promises to be very effective means of handling difficult machining operations on fatigue-critical steel parts.

### 9.2.3 Macchi MB-326H Fatigue (G.S. Jost, G. Clark and T.J. van Blaricum - AMRL)

The RAAF fleet of Macchi MB326H trainer aircraft is to remain in service until the year 2000. The loss of an aircraft in 1990 as a result of fatigue failure of a wing cast doubt as to whether this requirement could, in fact, be met.

Following the crash of Macchi aircraft A7-076 in which a wing was observed to have separated in flight, a substantial effort was initiated aimed at identifying and understanding the problem and then at verifying the current status and the future airworthiness of the remaining fleet. Following detailed examination of the crashed aircraft, in which the failure was found to have been caused by the presence of an undetected wing fatigue crack, teardown inspections of a further 11 high-life wings and a review of the wing fatigue damage model were carried out. These studies revealed that the structural integrity of many of the higher-life wings could not be substantiated, and it was concluded that most wings needed to be replaced at an early date—new wings have since been purchased from Aeronautica Macchi. The loss of aircraft A7-076 also focussed attention on the condition of other parts of the aircraft structure and AMRL is currently carrying out teardown inspections of flaps, ailerons, rudders and elevators to verify that these parts of the structure are safe for flight without the need for refurbishment.

Fuselage and tailplane teardown inspection at AMRL revealed two other areas of concern with the Macchi structure. Cracking was found in the fuselage centre section steel booms to which the wings attach and stress corrosion cracking was found in the tailplane spar caps.

#### 9.2.3.1 Fuselage Centre Section Boom Testing Program (T.J. van Blaricum - AMRL)

There are four regions of the centre section booms that are of interest. These are the fore and aft engine inlet ducting attachment screw holes, the fore and aft flange bolt holes, the web vertical bolt holes which retain magnesium spacer blocks which separate the upper and lower booms in the centre section and the wing attachment lugs.

The original aim of this testing was to establish, for the 04C booms fitted to Australian Macchi aircraft, that crack growth from the bolt holes was at a higher rate than cracking from screw holes. This needed to be confirmed as the screw holes are not inspectable, and safety-by-inspection for the centre-section is based on monitoring bolt hole cracking.

A total of four 04C tension booms have been tested using a modified FALSTAFF loading sequence developed at AMRL. This was designed to produce tension and compression loads on the boom which are closely representative of those experienced in service. The first two booms suffered attachment lug failure due to the presence of corrosion. The third boom also suffered a lug failure but due to the presence of a metallurgical defect. The fourth boom failed from a bolt hole cracking. All booms have yielded valuable flange bolt hole crack growth information which has been used in determining suitable inspection intervals for service aircraft. Two of the boom tests demonstrated that crack growth in regions where screw and vertical bolt holes almost

intersect can be at about the same rate as the crack growth from bolt holes. The tests have shown that wing attachment lug crack growth progresses at a much slower rate than for crack growth in bolt holes and these items can be easily inspected. With the exception of the vertical bolt hole screw hole intersection, screw hole crack growth is at a much lower rate than that for the flange bolt holes. Inspection intervals for the centre-section booms will continue to be based upon the requirements of those for the flange bolt holes.

#### 9.2.3.2 Tailplane Testing (T.J. van Blaricum - AMRL)

The tailplanes are to be refurbished by replacement of the corroded spar caps. Replacement will not be completed until late in 1995. To establish that the tailplanes are safe for flight in the interim period AMRL were requested to carry out static limit and ultimate load testing to verify that there was no significant reduction in residual strength. Two tests are required: one for the critical bending (Load Case D), the other for torsion (Load Case A).

Both tests have been completed. Load was introduced into the test article via bonded rubber tension patches, a conventional whiffle tree and hydraulic actuators. The AMRL test replicated the original certification test technique used by Aeronautica Macchi so that the results could be compared. The Load Case D test article failed at 250% of design limit load (DLL). Failure of the upper skin joint at the centre of the tailplane occurred at 191% of DLL. The final failure location was at the centre of the upper spar cap. Tear down inspection of the test article has revealed the presence of some stress corrosion cracking in the spar caps in the region of the attachment points. The Load Case A test was continued to 250% DLL without failure of any description.

### 9.2.4 Pilatus PC-9 Full-Scale Fatigue Test (J.M. Grandage - AMRL)

As noted in the previous ICAF Review, [5], AMRL is conducting this test with the aim of establishing a validated service life for the RAAF PC-9 fleet. Representative loading will be applied to the wing, empennage and fuselage. Load sequences on all components are currently (February 1995) being derived from flight strain sequences recorded from an instrumented aircraft flying representative sorties. Whilst most effort is being directed at deriving manoeuvre load sequences, buffet loads and ground loads are seen as being possibly significant (for the empennage and wing respectively) and will be represented on test. In parallel with the load sequence development work, test rig manufacture has been proceeding. Testing is expected to commence about August 1995. Several reports are in progress [10, 11, 12, 13].

### 9.2.5 Repair of Fatigue Cracking in Wing Skins of Lockheed C-141 Aircraft (R.A. Bartholomeusz - AMRL)

The Lockheed C-141 is a heavy-lift transport aircraft in operation with the USAF. The wing structure of this aircraft comprises integrally-machined 7075-T6 aluminium-alloy wing panels with risers, joined along longitudinal bolt splices. Fatigue cracking was found in the risers and in the wing skin originating from fuel-cell weep holes in the risers in the lower wing skin. After initiating in the risers, the cracks propagate in a direction normal to the applied load. Some of these cracks can be removed by reaming the holes, but many cracks are too large for this treatment. Due to the difficulties and cost of applying a mechanical repair to this problem a bonded composite repair was investigated as a possible solution. The repair materials selected for this application were boron/epoxy pre-preg tape for the composite patch and a structural film epoxy for the adhesive. These repairs have been applied by a team from Composite Technology Incorporated, DynCorp and AMRL, and a separate team from the Warner Robbins Air Logistics Centre. The cracked weep holes were scattered throughout the wing structure and were occasionally very close to other fittings, pumps, ribs, wing splices etc. The design and application of these repairs had to be flexible to suit the range of repair locations. Each crack was repaired with either three or five boron/epoxy patches and, to date, more than 120 aircraft have been repaired with 466 individual patches, [14].

### 9.2.6 Helicopter Structural Fatigue (D.C. Lombardo - AMRL)

Helicopter structural fatigue work for the Australian Defence Force (ADF) has increased dramatically since the last review. Although most of the effort has concentrated on the Australian Regular Army (ARA) S-70A-9 Black Hawk, AMRL has provided support for almost all the helicopter types being operated by the ADF as well as advice on future procurements. A summary of the more significant tasks undertaken is given below.

#### 9.2.6.1 U.S. Army UH-60A and UH60L Black Hawk

Assessment of U.S. Army Structural Usage Monitoring (SUM) results

The U.S. Army, in conjunction with Sikorsky, performed a SUM program on three of its Black Hawks to determine how U.S. Army Black Hawks were flown and to compare this with the assumed design usage spectrum. Although many problems beset the program, useful data were obtained and analysed by Sikorsky. The results of this analysis were provided to the ADF. The ADF sent the results to AMRL with a request for examination and assessment for relevance to Australian operations of the Black Hawk and Seahawk. The results of AMRL's assessment, [15], have yielded interesting findings which have been communicated to the U.S. Army.

#### 9.2.6.2 Australian Regular Army S-70A-9 Black Hawk

Assessment of new usage spectrum

As reported in the previous review, [5], Sikorsky had been contracted to replace the original, U.S. Army usage spectrum for the Black Hawk with one based on Australian operations. Sikorsky has now provided the new spectrum to the ADF who passed it to AMRL for assessment. Several inconsistencies were found but, overall, AMRL agreed that the new spectrum better represented Black Hawk operations than the original usage spectrum. The overall result of the new spectrum is that component retirement times will be drastically reduced. Sikorsky has determined the impact of the new spectrum by using it to estimate the retirement times for five representative dynamic components. The revised retirement lives are approximately half the lives currently set for the components. However, the revised lives are preliminary only and negotiations are currently under way to proceed with the next phase of the contract in which Sikorsky will produce definitive retirement lives for 16 dynamic components.

Risk assessment of overflying preliminary revised component retirement times

As mentioned above, Sikorsky estimated revised retirement lives for five components, based on the new Australian usage spectrum. For two of the five components, the revised lives are approximately 1000 hours. As most of the Black Hawk fleet is close to or beyond the 1000 hour mark the ADF was concerned about the risks involved in flying these components until Sikorsky produces definitive retirement lives (expected in mid- to late-1995). AMRL quantified both the relative and absolute risks involved in overflying the preliminary new retirement lives. Essentially, the risk analysis concentrated on the increased levels of risk associated with accepting reduced safety margins on assumed material properties. The reduction in risk associated with restricting the flight envelope was also examined.

Fast Roping and Rappelling Device (FRRD)

The ADF has tasked DSTO Salisbury with designing and manufacturing an FRRD for the Black Hawk. The FRRD will provide a rapid means for up to ten troops to exit simultaneously from the Black Hawk while it hovers several metres above the ground. The FRRD is particularly suited for areas where there are no suitable landing sites. AMRL has provided, and will continue to provide, advice on the fatigue implications that the FRRD may have for the airframe.

In-flight measurement program

A majority of the Black Hawks in the fleet is experiencing fatigue cracking in the left-hand side inner-skin panel between frames FS295 and FS308 (U.S. Army and U.S. Air Force Black Hawks are suffering similar damage). These frames are located just forward of the side doors and carry the lift loads for the forward portion of the airframe. The ARA has asked AMRL to determine the source of the fatigue-damaging loading and a Black Hawk has recently been appropriately instrumented. Flight trials are to begin shortly.

#### 9.2.6.3 Eurocopter AS-350B Squirrel

ADF requirement to operate Squirrel above maximum certificated gross weight

Both the Royal Australian Navy (RAN) and the ARA operate the Squirrel and both desire to increase its operational efficiency by operating at up to 2100 kg gross weight. Since the Squirrel is currently certificated for a maximum gross weight of 1950 kg, the ADF came to AMRL for assistance on the implications of this change. AMRL provided advice which assisted the ADF in reducing the costs of the changeover. Ultimately, strengthened rotor-system components will need to be installed in the Squirrels but, by implementing a simple AMRL-derived algorithm, the ADF was able to avoid having to replace existing rotor-system components immediately.

#### 9.2.6.4 Bell OH-58A Kiowa

Installation of Forward Looking Infra-Red (FLIR) night vision system.

The ARA had an urgent need to equip its OH-58 light observation helicopters with a night vision capability. They already had the FLIR units and the onboard systems were in place. All that remained was to mount the FLIR to the fuselage. The ARA had designed a mounting system for the FLIR and sent it AMRL for assessment. AMRL determined that the design was inadequate and proposed modifications which have since been incorporated. The ARA now has an airborne night vision capability.

#### 9.2.6.5 Helicopter Procurement Programs

The ADF is currently in the middle of defining its requirements for two major helicopter purchases. These are: AIR87, which is examining the possibility of replacing the OH-58 Kiowa and UH-1H Iroquois with a single armed reconnaissance helicopter; and SEA1411/SEA1427 which will procure helicopters for the ANZAC frigates and off-shore patrol vehicles. AMRL has provided advice regarding structural fatigue issues to be considered during the procurement process.

## 9.3 FATIGUE OF CIVIL AIRCRAFT

### 9.3.1 Piper Navajo and Chieftain Splice Plates (S. Swift - CAA)

In January 1994, during an overhaul of a Navajo wing, it was noted that the splice plate joining the lower spar caps was cracked. Investigation of the splice plate confirmed fatigue, caused by operational loads and aggravated by residual tensile stresses and fretting, Fig. 2.

The CAA issued an Airworthiness Directive requiring a visual inspection of the splice plates until the problem could be investigated more thoroughly. However, it was soon obvious that safety-by-inspection would not be practical. For example, the cracks start on the inaccessible top surface of the splice plate and are not visible *in situ* (even using ultrasonics) before they exceed a critical depth. To allow inspection, frequent disconnection of the wings would be required (which is impractical).

The CAA was therefore left with no choice but to place a retirement life on the splice plates.

### 9.3.2 Nord 298 / 262 Fin and Tailplane (S. Swift - CAA)

For several years, the continuing airworthiness of Nord 298 aircraft in Australia has been dependent upon a rigorous Supplementary Structural Inspection Program (SSIP). The development of that SSIP by an Australian consultant engineer, with the cooperation of Aerospatiale, was reported in the 1991 Australian ICAF Review.

The SSIP has been, and continues to be, instrumental in discovering serious defects and incipient catastrophes. This year two serious defects were discovered—a fin attachment forging was found to be badly cracked at 27300 hrs TIS and a horizontal stabiliser-to-fuselage support stay was cracked at 23300 hrs TIS. Even robust designs such as the Nord 298 will crack sooner or later. These defects would most likely not have been detected as part of the normal maintenance schedule, thus confirming the value of the SSIP.

### 9.3.3 Bell 214ST Tailboom Cracking (S. Swift - CAA)

In September 1994, cracking extending over more than one metre was found between body stations 243 and 280 on the underside of a Bell 214ST tailboom. Additional cracking was also found in adjacent bulkheads, stringers and in an inspection panel. The area of cracking was almost entirely covered by the external emergency tail float cover, Fig. 3.

Following this report, the remainder of Australia's fleet were promptly inspected. The result: 60% were found to have some cracking in this area.

This is only the latest of several reports of major fatigue cracks in Australian helicopter tailbooms. In March 1991 a 340 mm crack was found in the tailboom of an Aerospatiale Dauphin AS365 (80% of the Australian fleet were later found to be cracked) and in May 1992 a 250 mm crack was found in the tailboom of a Bell 212.

Having discounted maintenance neglect as a cause, the frequency of tailboom cracking brings into question the adequacy of the methods used to show compliance with the fatigue certification requirements. It seems that the fatigue design of helicopter fins and tailbooms warrants closer attention.

### 9.3.4 Cessna 441 Fuselage Frames (S. Swift - CAA)

Three cases of cracking in high-time Cessna 441 fuselages have recently been reported—two cases around the emergency door seal, which may lead to loss of cabin pressure; and a broken bulkhead just forward of the wing.

The Cessna 441 was designed to the fail-safe principle, with multiple load paths for primary structure. Several cyclic pressure tests of the cabin structure were completed. The test and service experience confirm the achievement of the fail safe design objective.

### 9.3.5 Bell 206 Landing Gear (S. Swift - CAA)

Bell has reported numerous failures of landing gear cross tubes, Fig. 4, some causing serious injury as the helicopter rolled over and the main rotor blades struck the ground. Bell has recommended cross tube replacement, given that cracks can remain hidden under the support. By the time the cracks emerge, crack growth is rapid and design strength is soon lost.

This is another example of the dichotomy facing the fatigue practitioner—if cracking can't be reliably detected by practical means, part replacement is the only safe solution.

### 9.3.6 GFRP Glider Wing Fatigue Test (C.A. Patching & L.A. Wood - RMIT)

Progress on this test program has been reported in previous ICAF Reviews. The previous Review, [5], provided a detailed account of major incidents during the life of the fatigue test.

The test specimen was a complete Schemmp-Hirth 'Janus B' tip-to-tip wing assembly. The starboard wing was badly damaged in a major accident and was fully repaired using a variety of techniques. The port wing was purchased new from the factory. AMRL, the Gliding Federation of Australia and the Australian CAA have assisted in the test program.

Fatigue testing has stopped after a total of 35 482 simulated flying hours. The adequacy of the design of repairs and the procedures followed during the application of the repairs has been adequately demonstrated. Subsequent significant failures were found only in the GFRP root rib in regions of concentrated load transfer. The material in these areas had, however, previously been subjected to high loads during a crash landing. The new replacement rib had reached only 7 135 hours when testing stopped. Further testing is unlikely because of budgetary constraints, [16].

### 9.3.7 Fatigue Life Assessment of the IS 28 B2 Sailplane (J. Ritchie & N. Mileshkin - RMIT)

The fatigue life of the IS28B2 sailplane has been assessed by making a comprehensive analysis of the fatigue performance of the structure under an actual load history. This has consisted of a teardown inspection of an unserviceable IS28B2 airframe, in conjunction with construction drawings, to identify fatigue-critical areas, the attachment of electrical resistance strain gauges in these areas and the determination, in association with the theoretical analysis, of the stress/g during various flight roles, the measurement of g load spectra using fatigue meters in three IS28B2 sailplanes, and a one-year survey of flight operations to identify the distribution of certain roles and manoeuvres amongst the fleet.

With this information, fatigue life calculations according to current practice were made using well-established fatigue data representative of each critical area, but adjusted to give fatigue performance agreeing with results of a full-scale test conducted by the manufacturer.

Since all research was conducted at the Gliding Club of Victoria on their fleet of IS28B2s, the comprehensive data base and life estimates are specific to local operations.

## 9.4 FATIGUE -RELATED RESEARCH PROGRAMS

### 9.4.1 Crack Tip Strain Determinations (J.M. Finney - AMRL)

In recent years a system of experimentally determining strains in crack tip regions using 25 μm microgrids has been developed. The system comprises a photoresist duplication of a master grid onto a metal specimen and photographing through a microscope the grid under load in a test machine. The enlarged grid is further magnified in a separate microscope and digitised using an attached CCD camera, thus measuring displacements and allowing the determination of strains. The present system is accurate only for plastic strains and a development is underway to enable elastic strain determinations. It is hoped that the elastic displacements, measured over a given region, will enable the experimental determination of the crack tip stress intensity factor, and the intention is to apply the technique specifically to the situation of multiple cracks in arbitrary arrays. This approach utilises the numerical basis described below. The ultimate aim is to develop a damage tolerance approach to situations of multiple cracks.

### 9.4.2 Determination of Stress Intensity Factors from Crack-Tip Displacement Fields (M. Heller & M. Richmond - AMRL)

It is planned to use the measured displacements of approximately 50 points (ie micro-grid intersections) near a crack-tip, Section 9.4.1, to determine the crack tip stress intensity factor for complex geometric and/or loading conditions.

The approach taken is to make use of a stress function written in terms of a series expansion, appropriate to a crack subjected to arbitrary remote loading. Using this stress function and standard equations of elasticity, the relationship between near-tip displacements and the stress intensity factor can be obtained, in terms of the unknown stress function coefficients. Hence, in principle, given measured displacements, the value of K can be determined.

Numerical work to date has successfully implemented this approach in a computer program which solves for the unknown coefficients simultaneously, when given a set of theoretically determined displacements. The K for a standard test case was accurately obtained by this approach. In typical test cases where data more remote from the crack-tip must be used, more coefficients are required to achieve accurate estimates of K.

The next stage of work will involve the implementation and numerical testing of a least-squares formulation to obtain the unknown coefficients in the presence of error in the displacements. The significance of the following variables on the quality of the determined K will be specifically addressed: (i) number of coefficients, (ii) distance from the crack tip and (iii) error levels in the displacement measurements. Following this it is planned to undertake experiments, including the testing of plates containing multiple cracks.

### 9.4.3 Fatigue Crack Growth Model for Short Cracks based on Crack Closure (Y.C. Lam - MU)

A crack growth model based on crack closure has been developed to predict short crack growth behaviour under constant amplitude loading. The model is able to explain the higher growth rate of short cracks when compared with that of long cracks. The predictions of the model agree well with experimental observations in the literature, [17].

### 9.4.4 Secondary Bending in Joints (R.L. Evans - AMRL)

Secondary bending often occurs in structural joints, and to assess its influence on the fatigue life enhancement technique of hole cold-expansion, an experimental program was undertaken, [18]. The specimens were similar to those of a UK secondary-bending joint design, [19], and were fabricated from an aluminium alloy common in the F/A-18 aircraft. Approximately half of the specimens were chemically coated and had sealant applied between the components, while the remainder were assembled without these surface treatments. All of the specimens were assembled with hole cold expansions of either zero or 4%, and fastener interferences of either zero (neat fit) or 1.1%. The fatigue tests were carried out under an F/A-18 manoeuvre loading program at two maximum net-area stress levels. The results indicate that the effect of cold working is substantially reduced in specimens with secondary bending. For any marked extension in fatigue life, additional treatments, for example, the combination of cold expansion and interference fit, are required. The application of sealant between the components of the specimens did not improve the average fatigue life of the specimens. Prior to testing, it was believed that the sealant would reduce the amount of fretting and hence improve the fatigue life. The amount of surface fretting was reduced, but it appears that the load transfer through friction was also reduced, which would adversely affect the fatigue life of the specimens with sealant. The resultant fatigue life of these specimens was unchanged.

A two-dimensional finite-element (FE) elastic analysis of a secondary-bending bolted joint was conducted, [20], to provide information relevant to the secondary-bending fatigue testing program mentioned above. Three different approaches were employed to model the bolt/plate interface and the results were compared with thermoelastic stress measurements. This work indicated that the *general* load-transfer characteristics of the secondary-bending specimen could be obtained from the two-dimensional models. Figure 5 schematically shows the displacement of the FE model (gripped section excluded) under tensile load. The bending is due to the offset of the neutral axes of the different sections from the line of the applied axial load. Three-dimensional effects appear to dominate in the region of the bolt holes, and a three-dimensional representation would be needed to model the strains *in these regions*.

The experimental work on secondary-bending specimens has been merged with complementary experimental work on multi-layer joints which examines the main variables with respect to fatigue life, [21].

### 9.4.5 Fatigue Resistant Holes (A.K. Wong - AMRL)

A new cold-working process has been developed for manufacturing holes which have exceptional fatigue resistant properties. It is well known that cold expansion can greatly enhance the fatigue life of holes in a metallic component. In particular, the split-sleeve expansion process developed by Fatigue Technology Inc. (FTI) has been widely used and accepted within the aerospace industry. The new method differs from the FTI process in that it is an integral part of forming a hole rather than a process which is applied to an existing hole. As such, this process is seen as more appropriate as a manufacturing technique rather than as a repair technique. Despite this, the superior fatigue resistance of holes produced by this method, Figs. 6 and 7, and its simplicity and ease of adapting to existing aircraft manufacture tooling, make it a potentially valuable production technique. A patent is currently being sought.

### 9.4.6 Composite-to-Metal Mechanical Joints (S.C. Galea - AMRL)

The static and fatigue properties of composite-to-metal mechanically fastened joints of the configuration shown in Fig. 8 have been investigated with (a) various degrees of damage around the fastener hole in the composite coupon and/or (b) varying fastener/composite hole tolerances. In an attempt to simulate minor damage caused by incorrect hole drilling, a single delamination was introduced into the coupon prior to the drilling of the fastener holes. However, this damage was much less severe than the multi-delamination damage due to assembly, [22]—this damage scenario was simulated by impact damage using experimental techniques summarised in [23]. The impact damage, through which a fastener hole was subsequently drilled, was produced by a 4 Joule energy impact and constrained to a 20 mm window. Three damage conditions were investigated, viz., (i) damage at holes A and B in Fig. 8, (ii) damage at hole A and (iii) damage at hole B. Static tests of these mechanical joints,

using a press-fit fastener in the composite coupon, showed no significant degradation in static strength when the damage was introduced.

Composite-to-metal bolted joint specimens with various damage conditions and two fastener/composite hole tolerances, viz. press-fit (snug-fit) and a generous clearance-fit, were also subjected to sequence loading. The clearance-fit was nominally 0.04 mm (0.0015"). The load sequence used for this program was the McDonnell Douglas Aerospace (MDA) 300 hour block (MACSEQ). The peak loads applied to the coupons were derived from MDA wing fold strain (load) data and then all loads were increased by a factor of 1.12, giving a peak tensile load of 30 kN. The maximum compressive loads were approximately half the magnitude of the peak tensile loads. Several techniques for evaluating the health of the joint were investigated, viz. ultrasonic C-scanning, joint load/displacement hysteresis measurements, fastener rocking using shadow moiré fringe measurements and changes in load transfer using piezo film sensors, [24]. Results, to date, of the variation in joint fatigue lives due to the various test conditions are summarised in Fig. 9. As a comparison, this figure also contains results for the hot/wet case using a press-fit condition, [25]. Note that the failure mode of the joint was by fastener failure. Figure 9 shows that:

1) Initial impact damage around the composite holes appeared to cause an insignificant effect on the fatigue properties of the joint for the two types of fits investigated, viz. press-fit and clearance-fit.
2) The introduction of a clearance-fit appeared to significantly degrade the fatigue properties of the joint compared to the press-fit case.
3) The hot/wet and the high loading cases caused the greatest reduction in fatigue life of the composite-to-metal mechanically fastened joints, with fastener failure occurring at fatigue lives considerably below those observed at RT/ambient under the same load conditions.

A thermoelastic investigation, using the FAST (Focal-plane Array for Synchronous Thermography) system developed at AMRL, of the composite-to-metal bolted joints with various damage states and press-fit fasteners, [26], indicated that damage could be detected. However, it appeared that the damage only slightly altered the thermoelastic images. A bolted joint without damage and with a generous clearance-fit was also analysed and results show a dramatic increase in the amount of secondary bending and fastener rocking compared to joints with press-fit.

### 9.4.7 Analysis of Bonded Inserts for Improving the Fatigue Life of Holes (R.L. Evans - AMRL)

Preliminary work using constant amplitude loading, pertaining to a bonded insert in a cracked hole in a plate, has shown that the insert significantly increases the fatigue life of the specimen, [27]. However, further combined experimental work by AMRL and DRA in the UK using FALSTAFF demonstrated that the adhesive for bonding the insert failed prematurely under the peak loads of the spectrum. These failures prompted an AMRL elastic finite-element (FE) analysis to optimise the design of the bonded-insert system, a system which is aimed at reducing the maximum plate stress and keeping the adhesive stresses below acceptable limits, [28]. In this work the plate is aluminium alloy, the insert is steel, and the adhesive is either an epoxy or an acrylic. Three main geometric cases were examined, namely; Case 1: circular hole (with and without a crack) and sleeve insert, with different constant adhesive thicknesses and varying adhesive thicknesses; Case 2: extended hole (slot) and insert, with different constant adhesive thicknesses and amounts of extension, and Case 3: elliptical hole and insert, with adhesive thickness varying elliptically.

Cases 1 and 2 were found to be unsatisfactory. In all variations of Case 1 the adhesive stresses were too high, and when the adhesive stresses were significantly reduced in some variations of Case 2, the maximum plate stress was only 6% lower than for the case of the same geometry but without the insert. The displacement diagram for Case 2 with an adhesive thickness of 0.2 mm and an extension of 7.9 mm, is shown in Fig. 10. Stresses at three indicated points on the adhesive/plate interface for this case, are also listed in Fig. 10. The most satisfactory model, from a stress analysis point of view, was that of an elliptical hole (a:b = 1:4) and insert, with a maximum adhesive thickness of 2 mm and a minimum of 0.5 mm (Case 3). The stress concentration factor of the hole was reduced from 1.8 to 1.47 by the bonded insert; however, the adhesive stresses were slightly above yield in one region. It was concluded that to reduce the peel and shear adhesive stresses and the maximum plate stress, a bonded-insert system utilising a long, narrow slit and variable adhesive thickness is required, which for a fastener hole is not very practical.

### 9.4.8 Time- or Cycle-Dependent Crack Bridging (L.R.F. Rose - AMRL)

Solutions are presented in [29] to the problem of a Mode 1 crack that is shielded by bridging tractions that can decay with the passage of either load cycles or time. The problem contains two competing time- or cycle-dependent processes, namely, degradation of the shielding tractions and crack advance. Particular emphasis is placed on the roles of thresholds for crack advance or shielding degradation. It is shown that the problem can reduce to either elastic/perfectly plastic bridging tractions (fatigue crack growth applications) or to fracture in the presence of a viscous process zone (monotonic loading applications). Solutions are illustrated by specific application to bridging by linear springs that soften with passing load or time. The springs may represent the action of a repair patch bonded over a crack in a metal plate, or of bridging fibres in a metal-matrix composite or laminate.

### 9.4.9 Evolutionary Structural Optimisation (G.P. Steven, O.M. Querin & Y.M. Xie - U of S)

The Evolutionary Structural Optimisation (ESO) procedure developed by Steven and Xie has been used to determine the size and shape of repair patches for cracked metal aircraft skins. The analysis has been performed on classical crack models: single-edge crack, centre crack and loaded hole with cracks on both sides. The results are very promising and illustrate that the ESO procedure can be a very useful and powerful design tool, [30]. ESO has also been used to determine the optimum grind-out shape for corrosion pits in thin metal surfaces. The shapes deduced provided significantly reduced stresses associated with the pit, [31].

### 9.4.10 Bonded Composite Repair of Metallic Aircraft Components (A.A. Baker & R.J. Chester - AMRL)

The ability to predict the rate of fatigue crack growth in patched panels is an important requirement for certification of bonded composite repairs. To achieve this capability, at least for simple patching configurations, Rose's one-dimensional analytical model, [58], of patching efficiency was adapted to predict fatigue crack growth in patched panels and extended to allow for disbond growth in the patch system. This work was reported at the AGARD Structures and Materials Panel meeting in Seville in 1994, [14].

Preliminary experimental work indicates that this is a promising model for predicting crack-growth behaviour over a limited range of variables including: a) artificial disbonds, b) growing disbonds, c) adhesive thickness and d) test temperature.

Fatigue crack propagation tests were conducted on 2024 T3 specimens 3.14 mm thick having starting cracks about 5 mm long repaired with unidirectional boron/epoxy 7-ply patches 0.9 mm thick. The patches were bonded with adhesive FM 73 at 120°C, following surface treatment using the silane process. For comparison, similar tests were conducted on unpatched specimens.

A series of specimens were made with artificial disbonds (using teflon inserts) ranging from 10 mm to 60 mm. Tests were conducted at a peak stress of 138 MPa and R = 0.1. The crack-growth results, Fig. 11, show that, as expected, crack growth rate increases dramatically with increasing disbond size. Figure 12 plots crack length versus cycles at a peak stress of 138 MPa and R = 0.1 for several sets of panels with an adhesive thickness of approximately 0.15 mm. At this stress level disbond growth, shown inset, was significant in some specimens. In these tests significant disbond growth occurred at the patch/adhesive interface.

Figure 12 also shows, as solid lines, the predicted behaviour, based on the analysis outlined in [14]. The estimates for the disbond growth rates, indicated by the letters a to f, are based on the observed *maximum* disbond size observed in the tests. Thus the disbond growth rates assumed in producing the theoretical curves, although overestimates, are a reasonable approximation.

Further testing is required to validate this model over a wider range of variables, including other patching systems. Also, to complete the predictive capability, fatigue data are required on the rate of disbond growth in the adhesive/patch system.

Residual strength in the patched panel tests was shown to exceed material yield for the 2024T3, which is one of the main requirements suggested for certification of bonded composite repairs in the absence of data on the design ultimate loads for the structure.

### 9.4.11 Fatigue of Joints Representative of Bonded Repairs to Aircraft (P.D. Chalkley - AMRL)

A test program has been initiated to characterise the fatigue behaviour of joints representative of bonded fibre-reinforced repairs to metallic aircraft. The double-overlap joint has been chosen as the joint most able to reproduce the stress state present through a central section of a bonded repair. A compliance technique will be used to measure crack length in the joint. Crack growth rates will be plotted against an appropriate parameter characterising the stress state in the adhesive such as the stress intensity factor range and /or the adhesive shear strain range. The value of the exponent in the Paris equation for the fatigue of epoxy adhesives is known to be fairly high (about four or five). Consequently, threshold values for the onset of fatigue damage in the adhesive will be determined. The effect of hot/wet conditions on the fatigue threshold will also be evaluated.

### 9.4.12 Rapid Repair Methods for Aircraft Battle Damage (R.A. Bartholomeusz - AMRL)

The rapid repair of battle damage is a critical requirement for the wartime operation of military aircraft. A technology base is under development for the rapid repair of aircraft structure based on adhesive bonding and composite reinforcement. Mechanically-fastened, metallic patches have been compared to an equivalent repair based on adhesive-bonding combined with graphite fibre composite reinforcement. This work was carried out as part of a collaborative program with the RAAF and the USAF Aircraft Battle Damage Repair (ABDR) Programme Office. Tests underway included static strength and fatigue; although fatigue is only a minor factor in battle damage repair it provides an excellent test of repair performance. As prescribed by the USAF, the repairs were required to survive at least 100 hours of spectrum fatigue loading, withstand 80% Design Limit Load (DLL) in compression, withstand 150% DLL in tension and be simple to apply.

The metallic, mechanically fastened repair while being the quickest to apply, failed the 150% DLL tensile test because of poor load transfer into the repair. The bonded composite repair performed well in the fatigue study and the system was effective in reducing stress intensity at the crack tip so that crack growth could be reduced or stopped. The repair also met the USAF ABDR criteria and survived the test program with no evident damage or failure. These results indicate that further work in an effort to reduce the cure time and increase the shelf life of the resins used in this bonded-composite repair is warranted, [34].

### 9.4.13 Durability of Postbuckling Stiffened Fibre Composite Shear Panels (R.S. Thomson & M.L. Scott - CRC-AS)

An investigation employing theoretical and experimental techniques is being undertaken to assess the performance of thin stiffened composite panels in the postbuckling state. Tests were performed on three-blade-stiffened panels with overall dimensions of 340 x 340 mm and working dimensions of 250 x 250 mm. The integral blade-stiffeners were 25 mm high with run-out angles of 45°. The skins and stiffeners were co-cured in an autoclave using flexible tooling. The material used to manufacture the panels was T300/914C carbon fibre/epoxy, unidirectional, pre-impregnated tape. The skins were 1 mm in thickness, consisting of eight plies of tape, and the stiffeners were 2 mm in thickness, consisting of 16 plies. Panels have also been manufactured with a 50 mm diameter teflon insert positioned under the middle stiffener to simulate the presence of a delamination.

The first stage of the program consisted of static testing of the panels using a picture frame shear rig in a servo-hydraulic universal testing machine. Surface strains were measured using electrical resistance strain gauges while out-of-plane displacements were determined using linear variable differential transformers and shadow moiré interferometry techniques. A photograph of the out-of-plane displacement contours generated using the shadow moiré technique is shown in Fig. 13. The theoretical postbuckling response was calculated using the geometric nonlinear capabilities of the finite element package MSC/NASTRAN. A comparison of the results showed very good agreement between experiment and finite element analysis, [32].

In the second stage of the program, the performance of the panels under cyclic loading is being established, [33]. In tests conducted to date, it has been determined that undamaged panels can withstand cyclic loading in excess of two thirds of the static ultimate load. Further tests will be performed using specimens with teflon inserts where the delamination growth will be monitored during testing. In addition to the laboratory testing, several specimens are currently undergoing cyclic loading at an environmental test facility in North Queensland to assess the effect of adverse environments on the durability of the panels. In addition to mechanical loading, these specimens are being exposed to high humidity, ultra violet light, rain and relatively high temperatures.

### 9.4.14 Assessment of Flaw Severity from Fractography (G. Clark - AMRL)

The assessment of fatigue crack growth rates in parts under test, or in service failures, is usually based on quantitative fractography, in which distinctive progression marks are related to specific events in the aircraft history. However, obtaining the full crack length/aircraft flight history curve is extremely time-consuming and often extremely difficult. As a result of extensive experience acquired while assessing fatigue cracks in Macchi aircraft and in test articles, a simple method has been developed for assessing the relative fatigue severity of various locations in a component. This is based on several deductions made from the extensive teardown results in this program, principally that the overall extent of a fatigue crack is controlled by (a) the initial defect size and (b) growth controlled by the local stress field. Scatter caused by material and manufacturing factors influences fatigue mainly through the initial defect size, while geometrical variations influence the crack growth regime. Crack growth is assumed to occur in an exponential fashion such that log crack depth is linearly related to test history (e.g. number of flights): the experience gained analyzing many hundreds of cracks in fleet aircraft provides good support for this assumption. A further assumption, supported by observation, is that the slope of this log/linear plot of crack depth/ history effectively indicates the severity of stressing at the location of interest.

The usefulness of these observations is that, without having to exhaustively investigate each and every crack growth history, it is possible to make deductions concerning, for example, the relative severities of different crack sites on the basis of the slopes of very simple plots, using only the measured initial flaw size, and the final observed crack size; the sites associated with high slopes of the crack depth curve can be seen to be the most severely stressed. The approach has been used in to assess the relative severities of cracking at a number of hole locations in Macchi centre section booms.

While the observations made in the course of this program are fully consistent with what would be expected on a fracture mechanics/stressing basis, the advance made here is in acquiring confidence in the observations by use of a large teardown sample, to the extent that simple observations can be used to make major decisions on fatigue management, [35].

### 9.4.15 Improved NDE of Discrete and Distributed Damage (C.M. Scala & S. Burke - AMRL)

This research is directed at the development of improved techniques, based on ultrasonic and electromagnetic nondestructive evaluation (NDE), for the detection and assessment of discrete and distributed damage in RAAF aircraft components. The eddy current research is aimed at developing procedures for crack-depth measurement and for first- and second-layer corrosion quantification. The ultrasonics research includes the use of both conventional and laser-based ultrasonics, and is currently aimed at NDE for improved crack sizing, quantification of first-layer corrosion, and bond durability assessment of boron-epoxy overlays used in repair/reinforcement applications. Some recent highlights from this work are as follows:

*Detection of Distributed Cracking*

A novel method for detecting finely distributed surface-breaking fatigue cracks has been developed, [36]. The approach relies on the detection of anisotropy introduced into the surface by an oriented array of cracks and can be used for crack detection even when individual cracks are below the practical threshold for crack detection using conventional techniques. The feasibility of the technique was demonstrated on a specimen containing simulated cracking (in the form of electro-discharge machined slots) using (i) a directional eddy-current probe to detect anisotropy in the effective electrical conductivity and (ii) a line-focussed laser-ultrasonic source to detect anisotropy in the attenuation of ultrasonic Rayleigh waves.

*Crack Size Determination using Eddy-Current NDE*

A first-principles method for crack size determination using eddy-current NDE is currently being developed. The method relies on the approximate solution to Maxwell's equations in the limit of small skin depth and has been implemented in the laboratory using a PC interfaced to an HP impedance analyser. The method has been used to determine the depth and opening of a series of long electro-discharge machined (EDM) slots, [37], in 2024 aluminium alloy plates with depths varying from 0.5 mm to 12 mm. In a more recent development, the method has been used to determine slot depth and opening for EDM slots hidden beneath boron/epoxy reinforcements and for EDM slots running the full length of a 10 mm diameter bolthole. The slot depths deduced from eddy-current measurements agree with the actual slot dimensions to within 10% or better. Work is in progress to test the performance of the sizing technique on a series of open and closed fatigue cracks in aluminium alloy test specimens.

### 9.4.16 NDE of Corrosion (G. Clark - AMRL)

The inevitability of the onset of corrosion, particularly in ageing aircraft, and the problems associated with its detection were summarized in the previous Review, [5]. The difficulty of NDI of multi-layer structures, the sometimes confusing effect of local geometric detail and the efficacy of common and novel NDI methods were discussed. Specimens representative of a large transport aircraft spar cap and web have now been prepared with flaws representative of corrosion defects of various geometries. These will be examined during the next year or so to establish the effectiveness of various NDI methods in detecting the simulated defects.

### 9.4.17 Quantification of Hidden Corrosion (C.M. Scala & S. Burke - AMRL)

A new one-sided ultrasonic technique using ultrasonic Lamb waves has been proposed to solve the commonly occurring problem of detecting hidden corrosion in thin metallic skin where the corrosion has originated at the interface between the skin and a second layer in an aircraft component. A successful demonstration of the technique has been reported, [38], in which the ultrasonic Lamb waves were laser-generated. More recent work has shown that conventional ultrasonic probes can also be used for the practical implementation of the technique. Current research is aimed at establishing a quantitative basis for the technique and in particular at establishing minimal detectability limits.

### 9.4.18 Materials Technology for Through-Life Support of Propulsion Systems (B.J. Wicks - AMRL)

A task has been set up to evaluate the performance and assess the deterioration of current and candidate high temperature materials used in gas turbine engine and transmission system components, and to develop materials-based methods for extending the operational lives and minimising the maintenance costs for these components. It is expected that the task will involve evaluation of a wide range of degradation processes including wear, fatigue, creep, impact, fretting fatigue, erosion and hot

corrosion, and will investigate methods of repair and property restitution, re-coating and the application of improved coatings. The primary focus of the task is to provide cost-effective and practical approaches for improving the performance and extending the lives of aircraft engine and transmission system components, including helicopter components. The task will also involve a response to short-term requests from the RAAF for advice on service problems when appropriate. Particular consideration will be given to F 404 and TF 30 engines, [39-52].

## 9.5 REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN NEW ZEALAND

### 9.5.1 Andover C Mk.1 (W.L. Price - DSE)

A Loads/Environment Spectrum Survey (L/ESS) has been completed, [53]. Fatigue damage rates derived from strain data were lower than those derived using the current counting accelerometer-based fatigue monitoring system. Statistical analysis methods were used to compare the damage calculated by the two methods and to determine the criteria for implementing further L/ESS activities, Fig. 14.

### 9.5.2 P-3K Orion (W.L. Price - DSE)

An evaluation of P3-K fatigue life confirmed RNZAF concerns that the wing structure had exceeded the cleared fatigue life. A wing change program has been approved. Safety-by-inspection procedures and manoeuvre restrictions have been recommended as a means of reducing the risk of encountering airworthiness problems before the program is completed, [54].

### 9.5.3 Aermacchi MB339CB (S. Ferguson - DSE)

Computer software has been developed to process data derived from the Aermacchi airborne strain counter fitted to all aircraft. Error detection and correction routines have been defined to assist automated manipulation of the data. MB339CB fatigue damage rates were initially higher than expected, but the introduction of full syllabus pilot training activities have ameliorated in-service damage rates.

### 9.5.4 Fatigue Analysis (P.J. Riddell - DSE)

Research aimed at forming an improved understanding of the mechanisms of fatigue crack growth has been based on a direct current potential drop technique. The method has provided the means to evaluate the influence of individual load cycles on fatigue crack growth and to clarify load sequence effects. Crack tip plasticity effects are much more complicated than current fatigue models indicate. Crack wall deformation and plastic zone advance both combine to influence crack growth behaviour, Fig. 15—the positive slope of the curves is due to the electrically equivalent crack length measuring changes in crack tip opening displacement as well as crack length. The crack growth increment measured for the overload cycle at (b) was twice that measured at (a) for an equivalent-sized cycle under constant amplitude loading, [55-57].

### 9.5.5 Structural Data Recorder (P.J. Riddell - DSE)

A recorder based on a networked system of microprocessors is being developed for structural data acquisition. The system is being produced to reduce the problems associated with installing L/ESS instrumentation in compact military aircraft.

### 9.5.6 A-4K Skyhawk (W.L. Price - DSE)

A L/ESS has been initiated to assess the remaining structural life of the RNZAF Skyhawk fleet. Design, fatigue test and fatigue data analysis are being reviewed to identify the structural zones of the A-4K which are likely to be fatigue-critical. The L/ESS will involve manoeuvre and mission analysis in addition to fatigue damage evaluation.

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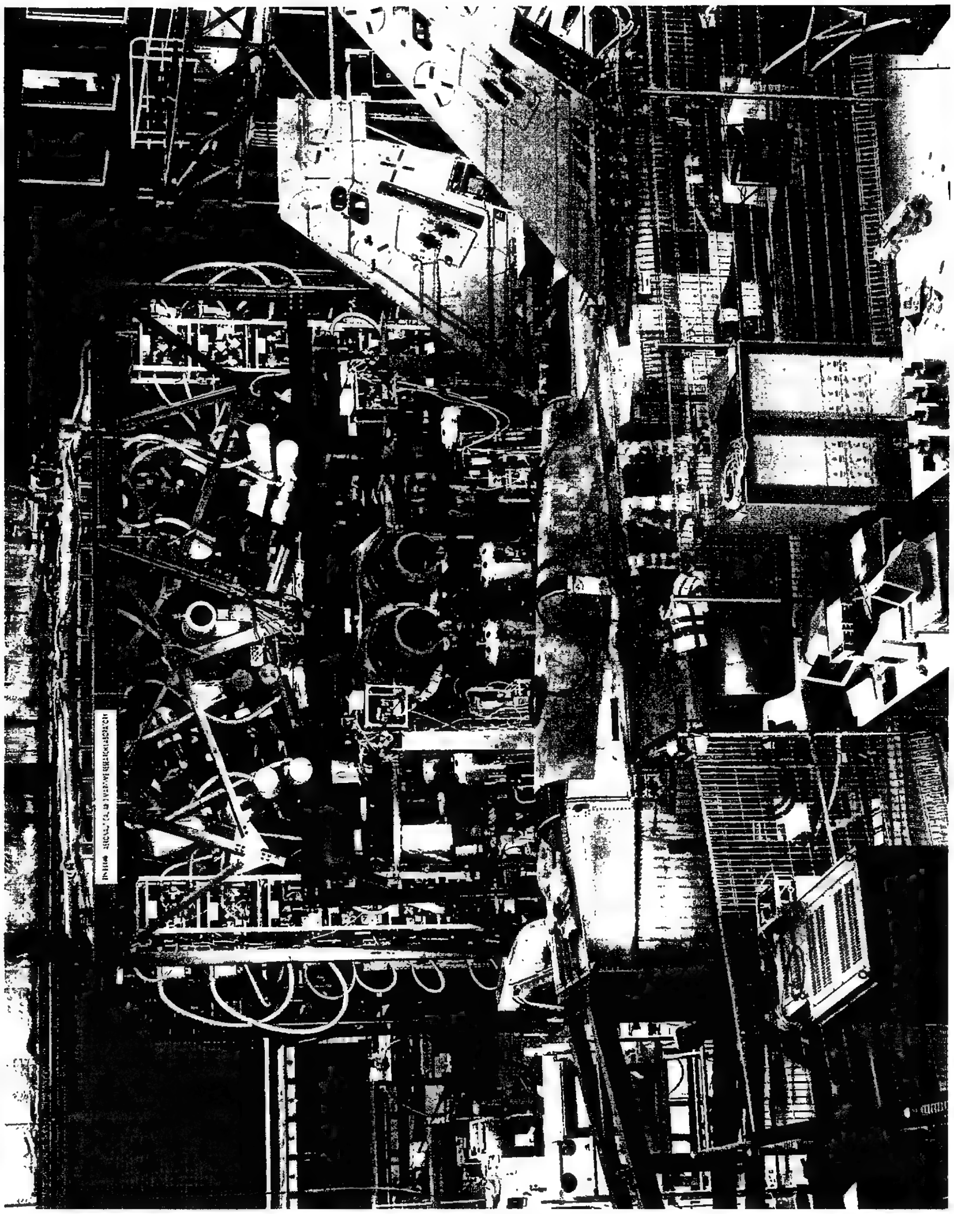


Fig. 1 F/A-18 Full-Scale Fatigue Test Rig, showing Dummy Installed Test Article
FT-46 Fatigue Test Article in Foreground

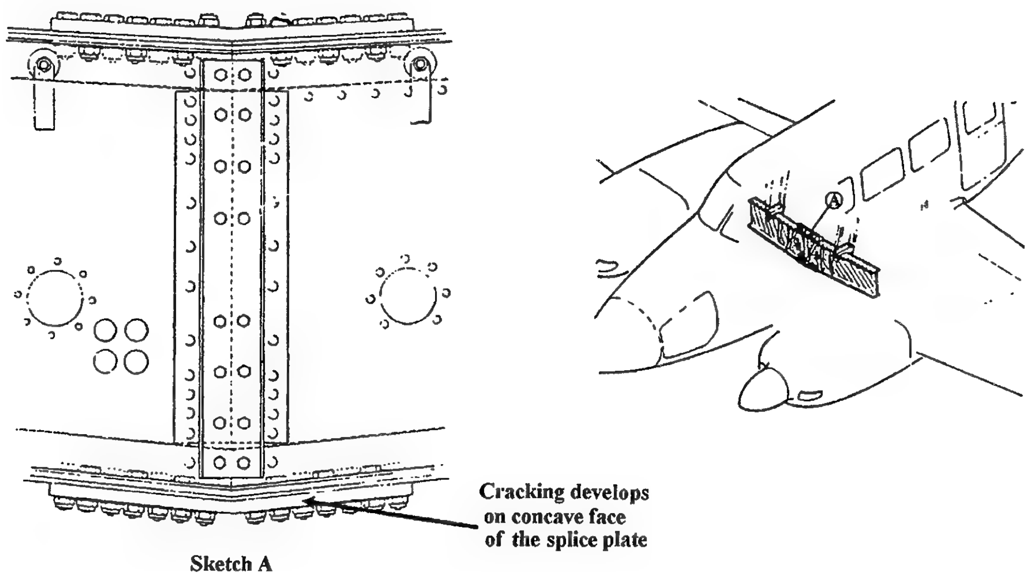



Fig. 2 Piper Navajo and Chieftain Splice Plates

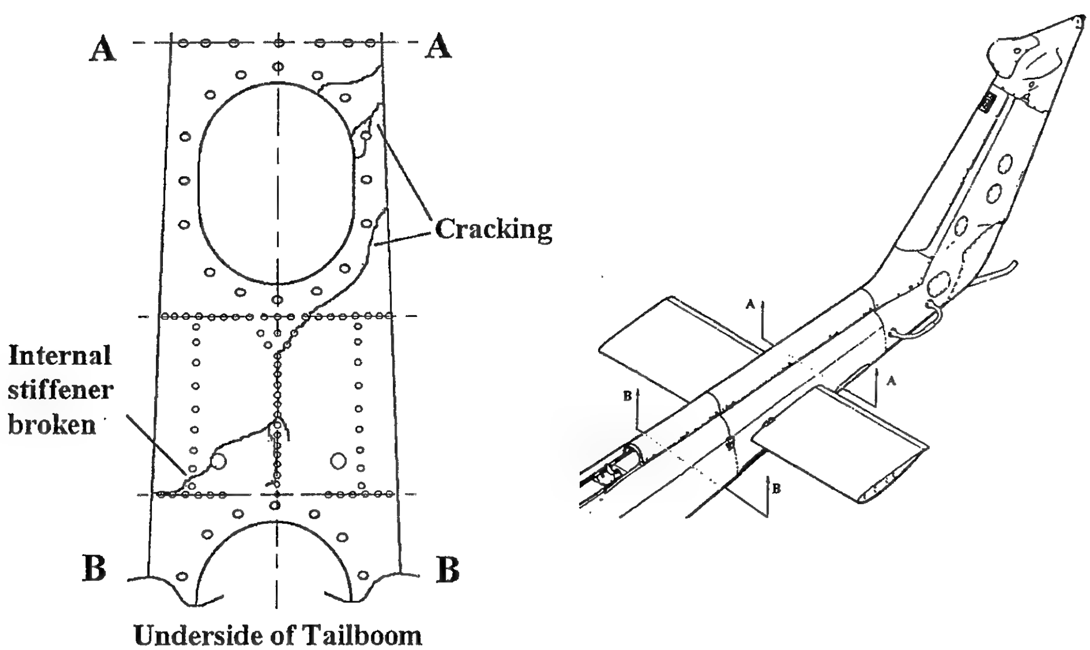



Fig. 3 Bell 214ST Tailboom Cracking

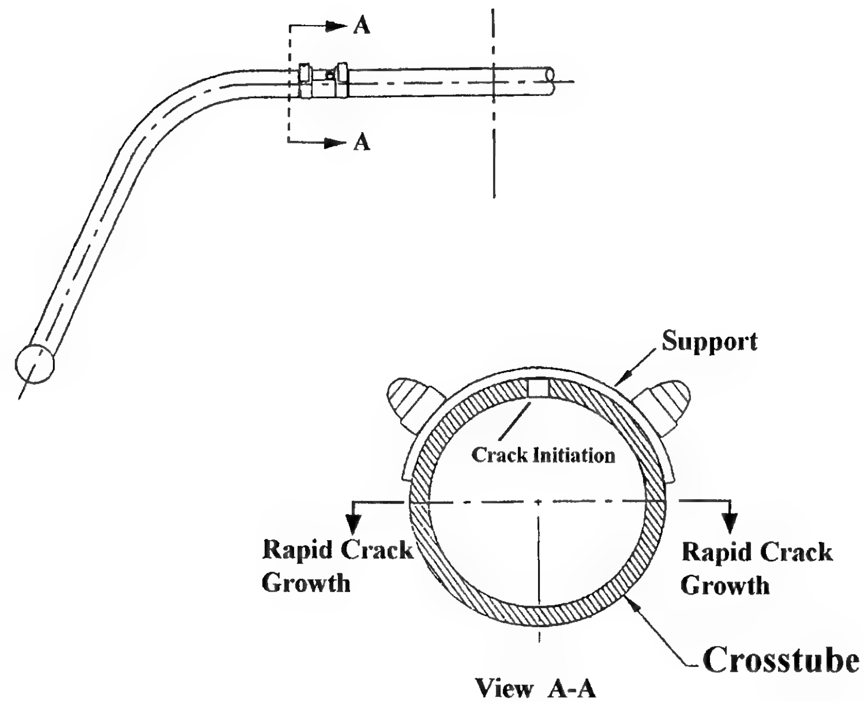
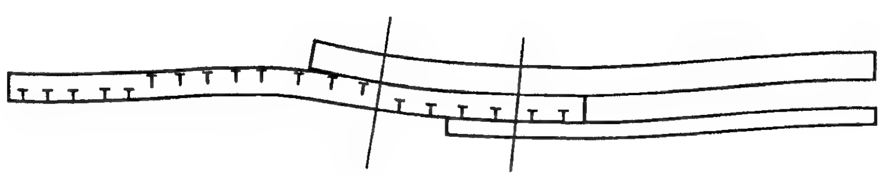



Fig. 4 Bell 206 Landing Gear



T = region of secondary tensile stress

Fig. 5 Schematic Displacement Diagram of Secondary-Bending Model

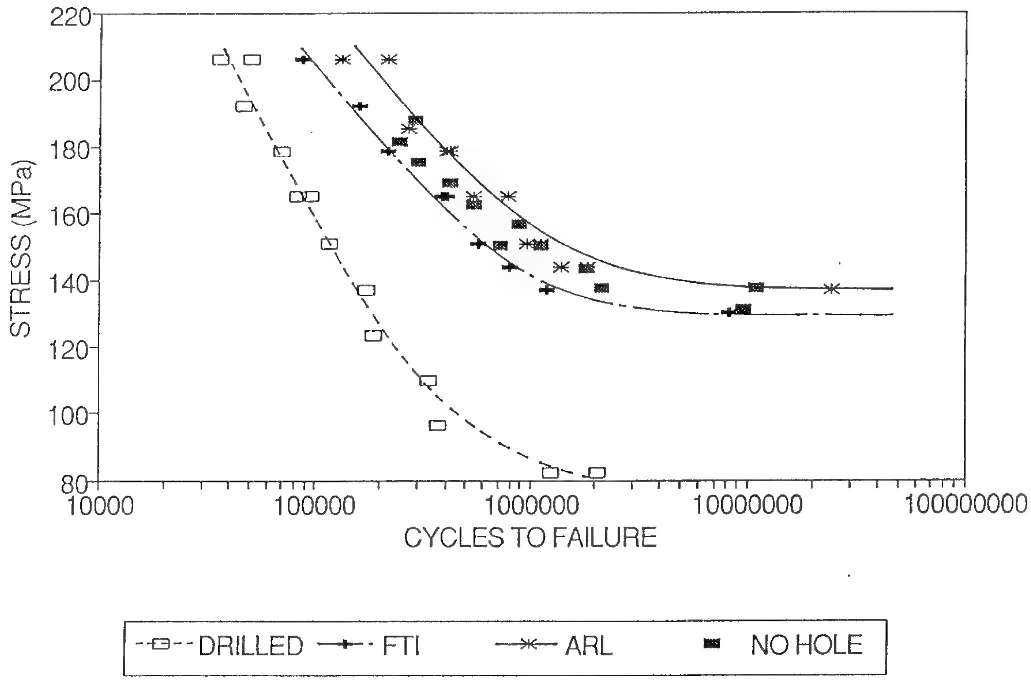



Fig. 6 Fatigue Resistant Holes - Results for Open Holes

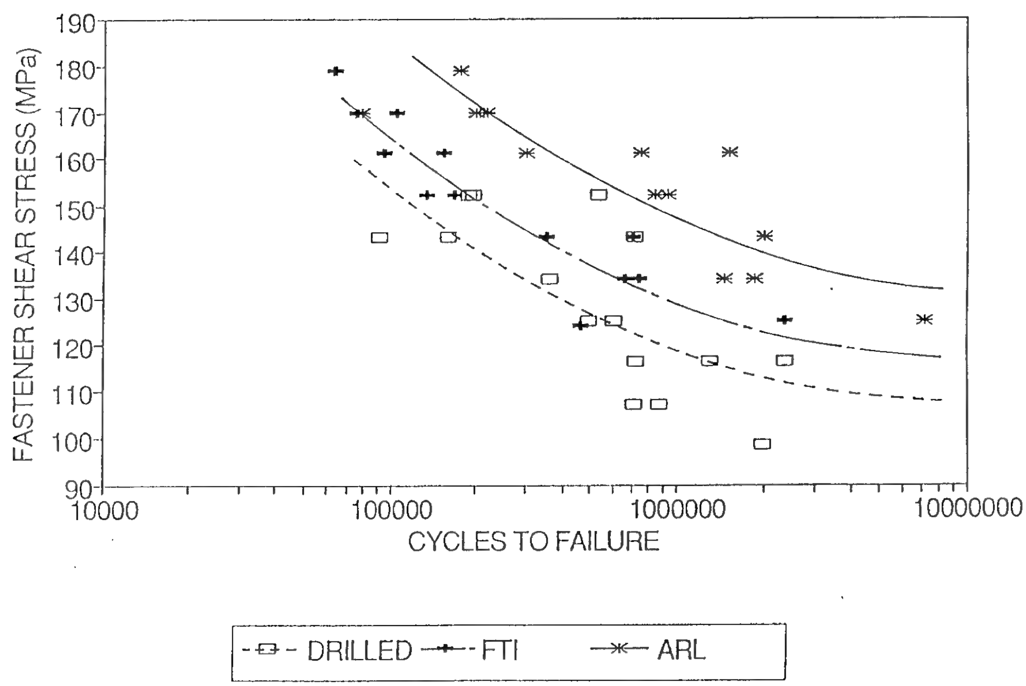



Fig. 7 Fatigue Resistant Holes - Results for Riveted Joints

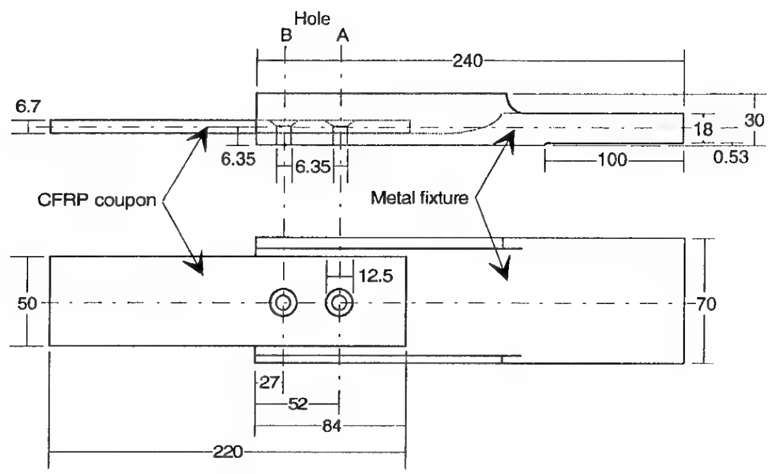



Fig. 8 The CFRP-to-metal bolted joint specimen

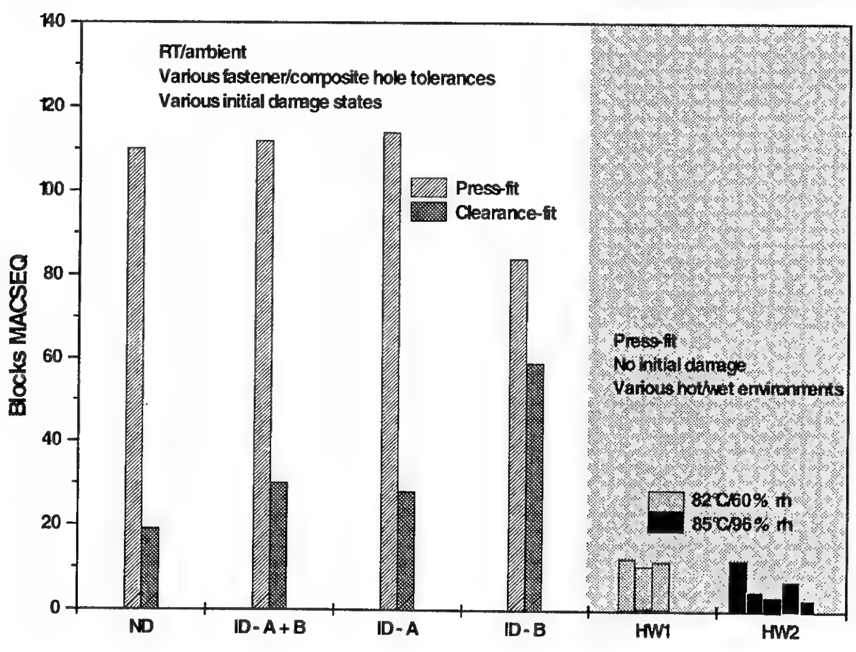
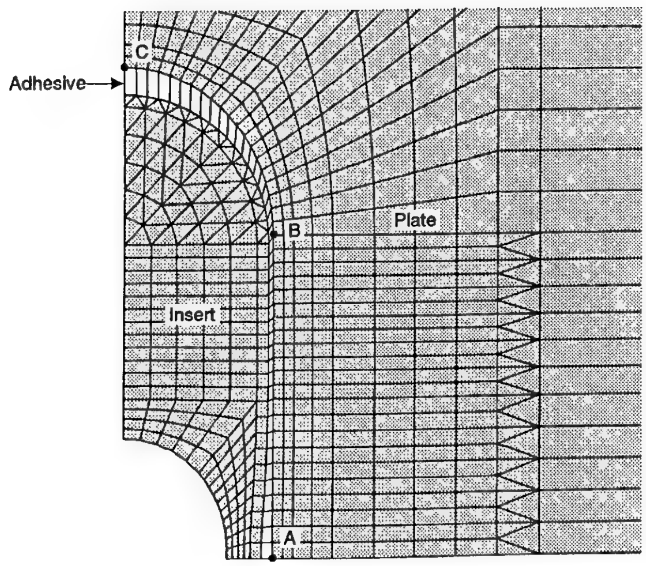



Fig. 9 Effects of test conditions on the fatigue performance of composite-to-metal bolted joints. Test conditions are: ND (no initial damage), ID-A+B (initial impact damage at holes A and B), ID-A (initial impact damage at hole A), ID-B (initial impact damage at hole B), HW1 (82°C/60%rh) and HW2 (85°C/96%rh)




Displacement diagram of insert region of Case 2

Stresses on the Adhesive/Plate Interface for Case 2

| Stress | Adhesive | | | Plate | | |
|---|---|---|---|---|---|---|
| (MPa) | A | B | C | A | B | C |
| $\sigma x$ | 14.4 | -13.2 | 28.7 | 14.4 | -12.9 | -66.8 |
| $\sigma y$ | 6.8 | -2.3 | 85.7 | 134.3 | 171.6 | 85.3 |
| $\tau xy$ | 0.0 | 11.4 | 0.0 | 0.0 | 9.2 | 0.0 |

Remote stress = 115 MPa

Locations A, B and C as per diagram above

Fig. 10 Result summary of Case 2 with adhesive thickness of 0.2 mm and extension of 7.9 mm.

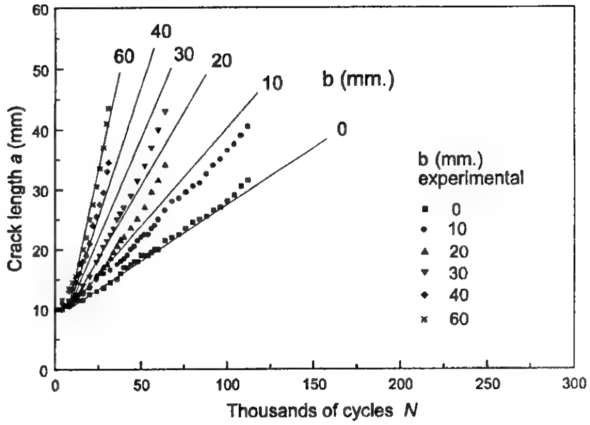
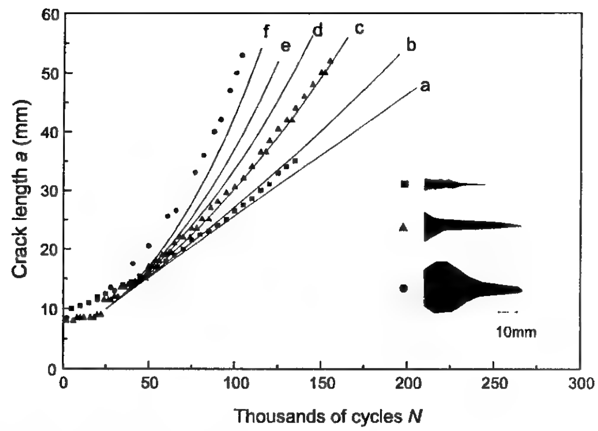



Fig. 11
Crack growth for patched specimens having artificial disbonds of length b. Solid lines are theoretical estimates.

Fig. 12
Crack growth for patched panels. Peak stress = 138 MPa, R = 0.1. Disbond shapes and sizes shown inset.



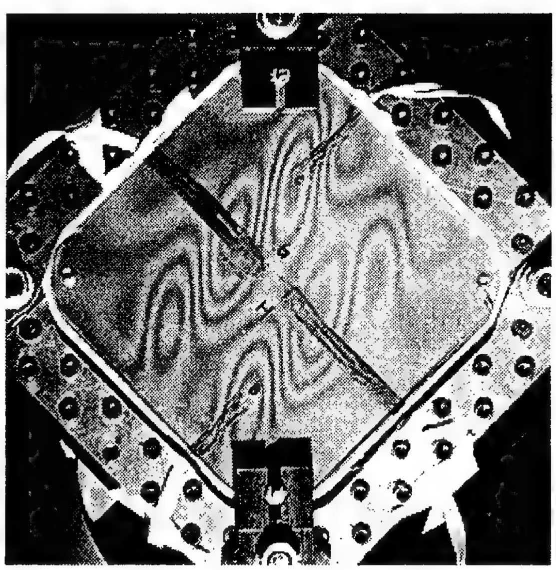



Fig. 13 Out-of-plane displacement contours showing postbuckled shape of composite shear panel

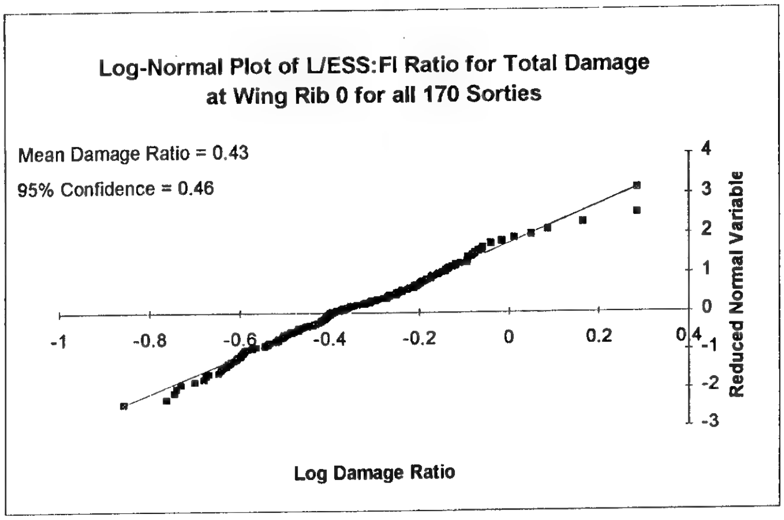



Fig. 14 Statistical analysis of Andover C. Mk. 1 L/ESS flight data

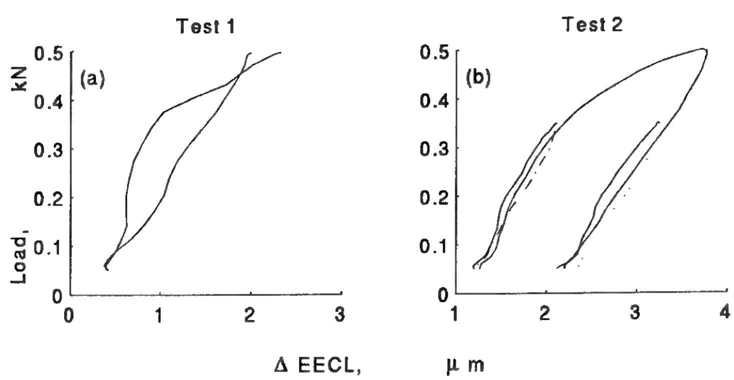



Fig. 15 Electrically equivalent crack length (EECL) variations with loading, measured on SENB specimens each with a 10 mm crack.

(a) single cycle from constant amplitude test
(b) overload cycle with two small cycles from overload sequence test consisting of 20 small cycles and one overload cycle.

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J.M. Grandage and G.S. Jost

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16. ABSTRACT

*This document was prepared for presentation to the 24th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Melbourne, Australia on May 1 and 2, 1995.*

*A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical and Maritime Research Laboratory, Universities, the Civil Aviation Authority and the Defence Scientific Establishment, New Zealand. The review summarises fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft.*

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